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SPACECRAFT POWER GENERATION

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National Aeronautics and Space Administration
Washington, D. C., U. S. A.

Presented to the Combustion and Propulsion Panel
of the Advisory Group for
Aeronautical Research and Development
at the California Institute of Technology

Pasadena, California
August 24-26, 1960

NATIONAL
AERONAUTICS AND SPACE ADMINISTRATION
WASHINGTON

Outline

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I. INTRODUCTION

This paper deals with the problem of generating electrical power in spacecraft outside the earth's atmosphere for mission durations of a few hours to several years. An estimate is made of the electrical power levels required in various spacecraft planned for development by the National Aeronautics and Space Administration, and then a discussion is presented of the research and development which is under way or which will be needed to provide adequate power from solar, chemical or nuclear energy sources.

II. ESTIMATED ELECTRICAL POWER REQUIREMENTS

Practically all space vehicles will require electrical power to operate communications equipment, instruments and tracking beacons. Some vehicles will also require power for operation of equipment for control of the internal environment, to control the vehicle attitude, to actuate a chemical or nuclear propulsion system or to power an electrical propulsion system. Spacecraft which land on the moon or planets will require power for most of the purposes mentioned. Figure 1 shows an estimate of the electrical power levels required in various spacecraft during the next few years. These data are intended to indicate the highest foreseeable power requirement in any given year for non-military spacecraft. Since the NASA plans to launch a total of 260 space vehicles during the next 10 years, there will be many vehicles flown in each year with power generating capacities lower than the maximum values indicated in Figure 1. Nearly all the vehicles which are scheduled through 1963 will require average power levels below about 250 electrical watts. Power for the unmanned vehicles can be provided by silicon solar cells in combination with nickel-cadmium

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batteries. Many of the scientific satellites launched by Scout or Delta vehicles will require less than 10 watts. An average power drain of around 20 watts is typical of the Pioneer V space probe and the TIROS satellites.

The one-man Mercury capsule will carry about 144 pounds of silver-zinc batteries to provide an average power of 70 watts while in orbit, with an emergency reserve. The peak power demand will be about 1000 watts. The Nimbus meteorological satellite is estimated to require an average power of about 250 watts. The Ranger spacecraft, to be launched by an Atlas-Agena B for lunar landing missions, is estimated to require about 115 watts of regulated power which will be provided by an oriented solar cell array capable of supplying 190 watts of unregulated power. The actual lunar landing package will require only 0.2 watt for a period of one to three months. The spacecraft for Venus and Mars probes will require about 300 to 500 watts.

Beginning in about 1965, the Saturn booster will be used to launch a three-man spacecraft which will be developed for the manned lunar circumnavigation mission called Project Apollo. It is estimated that the average power level required will be 1.2 kilowatts for 14 days or a total of 400 kw-hrs. A peak power demand of 4 kw is estimated.

During the late 1960's the Saturn may be used to establish an orbiting laboratory in connection with the manned lunar circumnavigation mission. The average power required for such a satellite may be about 1 to 3 kilowatts, with a peak power of 4 to 6 kilowatts and a mission duration of two weeks to two months.

Beginning in about 1965, it is estimated that experimental flight testing will be conducted with an electrical propulsion system consisting of an ion or plasma thrust device and a nuclear-electric power supply. A power level of either 30 or 60 kilowatts will be supplied by the SNAP-8 nuclear reactor, coupled with one or two 30 kw mercury turbogenerator systems. This power system is now under development jointly by the Atomic Energy Commission and the NASA.

Recently the U. S. Air Force initiated a contract with the Garrett Corporation's AiResearch Division for initial design and development effort of a 300 kilowatt nuclear-electric power system. The system is called SPUR, and it will utilize a high temperature liquid metal Rankine cycle.

Depending on the results of research now in progress on electrical thrust devices and on nuclear-electric power plants with turbogenerator or thermionic conversion systems, plans may be made by NASA for the development of electrical propulsion systems requiring power levels of one to twenty electrical megawatts. The primary application for such propulsion systems will be for planetary exploration missions.

III. POWER GENERATION SYSTEMS

A. General Survey

Electrical power may be generated in space from three basic energy sources: Chemical, solar or nuclear. Nuclear sources include nuclear fission reactors and radioisotopes. Figure 2 shows the variety of power conversion methods which may be used to generate electric power from these sources, with an indication of their state of development.

The only types of power supplies which have been used in satellites or space probes to date are photovoltaic solar cells and electrochemical batteries. Some of the more promising systems for future development are the combinations of solar heat collectors or nuclear reactors with mechanical, thermoelectric or thermionic power conversion. For space missions of a few days' duration, such as manned satellite missions, the use of fuel cells appears to be competitive with primary batteries and with chemical combustion auxiliary power units which employ mechanical power conversion.

In the following sections, each of the major types of power supplies are discussed, followed by a list of criteria for selection, and a discussion of the more promising types of power systems for each range of power levels. Several systems are not discussed because they either offer no potential of being competitive, or the research is still in too early a stage for proper evaluation. Examples include:

1. Thermal conversion by ferroelectric, thermomegnetic or pyroelectric processes.

2. Direct solar conversion by photoelectric (photoemissive) processes.

B. Chemical Power Systems

1. General Comments

Chemical power systems fall into two categories: Electrochemical and combustion types. The electrochemical systems provide direct conversion of chemical to electrical energy by means of batteries or fuel cells. The combustion types utilize the thermal energy of chemical reactions either by expansion of the reaction products through an open cycle converter or by heat transfer from the reaction products to a heat engine.

2. Batteries

At the present time, the available battery offering the highest energy per unit weight (about 50 to 70 watt-hours per pound) is the silver oxide-zinc primary battery. Unfortunately, gas evolution during recharging makes it impractical to seal the battery for use in spacecraft as a secondary rechargeable battery. The only rechargeable battery which has been used in space vehicles to date is the nickel-cadmium battery, which can be hermetically sealed. For multiple recharge useage, such as encountered with a solar cell power supply, the useable depth of discharge must be limited. For example, in a low altitude satellite orbit, about 6000 cycles are required per year, which at present requires that the depth of discharge be limited to about 10%. Therefore, only 1.5 watt-hours can be repetitively stored per pound of battery, compared to the 15 watt-hours per pound which are available during a single complete discharge. The useful capacity of any battery, of course, is influenced by the allowable charge and discharge time, the limits permitted on voltage regulation and the ambient temperature.

An investigation is under way to determine whether imperfect sealing of the nickel-cadmium batteries may have caused the determination of the power supply in the Pioneer V space probe. There is evidence from ground testing that paper separators between the electrodes tend to hydrolyze in the KOH electrolyte at ambient temperatures above 125°F ($\sim$50°C). Therefore, research is proceeding on improved separator materials and construction methods. Although development is under way to solve the mechanical, thermal and quality control problems of nickel-cadmium batteries, it is believed that more research on the electrochemical and corrosion problems is also needed. A battery with greater useable energy per unit weight would be very desirable. The silver-cadmium battery appears to offer some potential.

3. Fuel Cells

In order to obtain more watt-hours per pound than batteries can provide, a broad research and development effort is being conducted on fuel cells of various types. The hydrogen-oxygen fuel cell has a theoretical output of 1620 watt-hours per pound of reactants, which is the highest of any presently known practical electrochemical system. At 60% conversion efficiency, the fuel consumption is only one pound per kilowatt-hour. Electrodes are generally made of porous carbon or porous metal, with electrolyte (such as KOH solution) in the space between electrodes. A liquid electrolyte may be contained in a porous ceramic separator or a solid electrolyte may be used in the form of a membrane made of an ion-exchange resin. Hydrogen-oxygen fuel cells have operated under pressures of one to fifty atmospheres and temperatures from room temperature to 250°C. Some of the problems which must be solved before fuel cells can be used in spacecraft are:

a. Achieve long electrode life at current densities of about 100 milliamperes per square centimeter or higher.

b. Provide for separation and removal of the reaction product (water) and avoid mixing of gaseous fuels into a liquid electrolyte under zero-gravity conditions.

In order to use a fuel cell battery as a rechargeable energy storage device, provision must be made for storing the hydrogen and oxygen gas which is generated when the cell current is reversed or when a separate water electrolyzer is employed. The efficiency of energy storage with an electrolytically regenerative H_2-and O_2 fuel cell is presently below 40% because the voltage required for electrolysis is more than 2.5 times as high as the output voltage during discharge at a practical current density. The rather rapid decrease in output voltage with increasing current density will impose severe voltage regulation problems in most applications. The use of carbonaceous fuels such as alcohol with oxygen in fuel cells is complicated by the necessity of removing carbonate from the alkaline electrolyte.

It is believed that the two most promising applications for fuel cells in spacecraft are:

(a) The primary H_2 and O_2 fuel cell with cryogenic storage of the fuel and oxidizer for an output of 500 watts to several kilowatts and durations of several days.

(b) The electrolytically regenerative H_2 and O_2 fuel cell for energy storage in conjunction with solar power systems.

4. Open Cycle Combustion Type Auxiliary Power Units

Many types of auxiliary power units (APU's) have been developed for missile applications. Monopropellants such as hydrogen peroxide, ethylene oxide and hydrazine have been used with turbine expansion to drive an alternator. The APU in the X-15 aircraft is a hydrogen peroxide turbine system. Utilization of hydrogen and oxygen with excess hydrogen to limit the gas temperature is being applied with both turbine and reciprocating expansion engines. The low specific fuel consumption (below 2 lb/kw-hr) with H_2 and O_2, together with the possibility of using the liquid hydrogen to cool spacecraft components prior to combustion, makes this system attractive for manned spacecraft requiring several kilowatts for several days. Potential applications include the Dyna-Soar and the Apollo circumlunar spacecraft.

In order to avoid the temperature limit of about 1000°C imposed by turbine or reciprocating expansion engines, research is being conducted on magneto-fluid-dynamic (MFD) power conversion. Potentially, stoichiometric combustion could be used and an easily ionized material such as cesium added to the combustion products to produce an electrically conducting plasma, at a temperature near 3000°C, which is then expanded through a magnetic field to generate electrical current directly. The current is extracted through electrodes perpendicular to both the magnetic field and the plasma velocity vector.

The NASA is initiating a contract with Thompson Ramo Wooldridge, Inc., to investigate the processes involved in a generator (See Figure 3) involving inward vortex flow with an initially supersonic tangential plasma velocity. An axial magnetic field is imposed on the flow. This vortex geometry may be more desirable than the usual linear duct geometry, because of a reduction of volume of magnetic field required and because the velocities and mass flow rates may be varied and optimized independently. Much research on plasma dynamics and on electrode corrosion and ablation will be required before the potentialities of this system can be assessed. It is possible that the exhaust gases from an open cycle MFD generator could be regeneratively cooled and then expanded through a turbine to attain a higher conversion efficiency.

5. Closed Cycle Heat Engines with Combustion Heat

The use of chemical heat with a closed cycle heat engine such as a Rankine cycle turbogenerator system, a Stirling cycle reciprocating engine, or a thermionic or thermoelectric heat engine will permit the use of stoichiometric combustion and high flame temperatures, thereby producing combustion products at the maximum enthalpy. The U. S. Air Force is now supporting development of a five-kilowatt closed Rankine cycle system which uses stoichiometric combustion of hydrogen and oxygen as a heat source, and aluminum bromide as the working fluid.

The full utilization of the available enthalpy in the combustion products requires that the gases be cooled to a low temperature before being exhausted. In concept, the products of combustion could be passed successively through various heat engines, each optimized to give the maximum fraction of Carnot efficiency at each particular temperature of heat input. Studies are needed of the effect of cascading various heat engines on decreasing the fuel consumption in relation to the complexity, weight and unreliability which is added.

Recently the Radio Corporation of America and Thiokol Chemical Corporation tested a thermionic converter which generated 270 watts when heated by exhaust gases from a solid rocket. Corrosion of the cathode materials at temperatures of 2000°F to 3000°F by the combustion gases will tend to limit the operating life of such devices.

C. Solar Power Systems

1. Photovoltaic Systems

At present silicon solar cells of the p on n type are commercially available with conversion efficiencies of 8 to 12%. Figure 4 shows a sketch of a typical solar cell, which is one by two centimeters in area and about half a millimeter thick. It appears probable that an increase in the efficiency of single crystal silicon cells to 15% will be attained in the near future. Research is being conducted on improved single crystal materials such as gallium arsenide, which may permit efficient operation at higher temperatures than silicon. Research is also proceeding on growing large single crystals and depositing large area polycrystalline sheets on a suitable substrate, as well as depositing multiple layer p-n junctions which may utilize a wider band of the solar spectrum.

The development of solar cell arrays using available silicon cells is proceeding rapidly for application in a wide variety of space vehicles. Sun-oriented arrays with a total area of 20 to 100 square feet are being developed to generate average power levels up to 250 watts. For example, Figure 5 shows one of the 10 square foot solar cell panels under development by the Jet Propulsion Laboratory. Two of these panels will be used on the Ranger spacecraft for lunar missions. Each panel will carry 4340 silicon solar cells and generate 95 watts of power when oriented to face the sun outside the earth's atmosphere, at a temperature of 39°C.

When solar cells are used in earth satellites passing through the trapped radiation belts, consideration must be given to protecting the cells against proton and electron radiation which would reduce the power output by producing atomic displacements (holes) near the p-n junction and thereby reduce the minority carrier lifetime.

Preliminary results from radiation testing in electron and proton beams indicate that a one to two millimeter thickness of non-browning quartz or cerium glass will provide adequate protection over the solar cells for one year operation in the electron fluxes of the outer Van Allen belt (e.g., in a 24-hour orbit). Further ground testing and more accurate measurement of proton fluxes in the inner Van Allen belt are required in order to establish shielding requirements and life expectancy for satellites in the altitude range from 1000 to 5000 kilometers. Experiments by Rappaport and Loferski of the Radio Corporation of America have recently shown that solar cells made by diffusing an n-type layer onto a p-type material (as developed in the USSR) are considerably less sensitive to radiation damage than the p on n type, which are in large-scale production in the United States. The n on p type cells which were tested by RCA were produced under the direction of Mr. Joseph Mandelkorn at the U. S. Army Signal Research and Development Laboratories in Fort Monmouth, N. J.

Solar cell arrays which are designed to withstand launch vehicle vibrations and accelerations and to unfold and be sun-oreinted presently weigh from one to two pounds per square foot. Since the solar cells can produce about 10 watts per square foot, the system can deliver about 5 to 10 watts per pound when fully oriented. When the weight of nickel-cadmium storage batteries and the added weight of array to charge the batteries is added, the over-all power system for a low orbit satellite is likely to deliver only 2 to 3 watts per pound.

2. Solar-heated Systems

a. Solar Collectors

Solar heat collectors of either the flat plate or radiation concentrating types will be capable of generating higher temperatures in the vacuum of space than on the earth because convection heat losses are eliminated. Therefore, solar-heated spacecraft power systems appear quite promising. Flat plate receivers with surface coatings having a high ratio of absorptivity for the solar spectrum to emissivity in the infrared should be capable of attaining temperatures up to about 400°C in space, which is adequate to operate a low efficiency thermoelectric converter. Light weight solar concentrators of various types are being developed to provide high temperature heat sources for more efficient heat engines. For example, Figure 6 shows a 10 foot diameter umbrella type paraboloidal collector made of 1/4 mil thick Mylar plastic which has a vapor-deposited aluminum reflective coating of approximately 2000 Angstroms thickness on its front surface. The weight of this collector is about 0.3 pounds per square foot of frontal area. By reducing the weight of the hub structure, a weight of less than 0.2 pounds per square foot may be attainable. Figure 7 shows a 3 foot diameter electroformed nickel mirror with plastic foam backing which weighs 0.4 pounds per square foot. Figure 8 shows a 10 foot diameter foldable petal-type collector made of 1/4 inch thick aluminum honeycomb which weighs about 0.25 pounds per square foot of frontal area. Development of Fresnel mirrors and of inflatable plastic mirrors is also being conducted.

The temperature attainable in a heat receiver depends primarily on the radiation concentration ratio (the ratio of collector frontal area to receiver aperture area). The minimum acceptable aperture area is dictated by the angular accuracy of the reflector surface at every point, the degree to which specular instead of diffuse reflection is attained, and the angular accuracy of orientation toward the sun. Typically, the efficiency of a solar collector may be about 40 to 50%.

b. Thermal Energy Storage

Although solar concentrators permit the attainment of a high temperature heat source to operate a heat engine, provision must often be made for power generation while in the shade. As previously mentioned, the energy storage capacity of batteries is reduced by the need for repetitive sycling. Furthermore, if electrochemical storage were used in a satellite vehicle, mechanical conversion systems would stop and have to restart once during each orbit, and thermionic or thermoelectric converters would be exposed to severe thermal cycling. These factors lead to the interest in using thermal energy storage to operate a heat engine continuously while passing through the shadow of the earth or of a celestial body, instead of using electrochemical energy storage. A promising technique is to store energy while in sunlight as the heat of fusion of a material whose melting point is somewhat higher than the desired heat input temperature to the heat engine. While in the shade, the material would be allowed to freeze, releasing the heat of fusion to operate the engine at constant output. A high value for the heat of fusion is desirable. Figure 9 shows the heat of fusion vs. melting point for several materials which may be considered for thermal energy storage. It is seen that lithium hydride is very desirable for heat engines operating below its melting point of 682°C (1260°F). Sodium fluoride may be considered for use with boiling liquid metal cycles (such as rubidium or potassium) or Brayton gas turbine cycles at temperatures up to 992°C. The beryllium-silicon eutectic appears suitable for cycle temperatures up to 1089°C. Molten beryllium and particularly silicon would be desirable for use with thermionic converters having cathode temperatures below 1350°C or 1420°C, respectively.

The selection of thermal storage materials, and materials for their containment, will require a concerted research effort on high temperature metallurgy, including the investigation of phase diagrams, corrosion and diffusion rates, as well as physical, thermal and mechanical properties.

The materials shown in Figure 9 are representative of some promising candidates, although other materials such as various oxides or halides should also be studied. Research is also needed to exploit the heat of reversible chemical reactions for thermal energy storage.

c. Solar-Mechanical Systems

Three types of solar-mechanical power systems are presently under development: A 3 kilowatt output mercury vapor turbine system, a 3.8 kilowatt Stirling engine system, and a 15 kilowatt rubidium vapor turbine system. The first two will utilize molten lithium hydride for thermal energy storage. The rubidium cycle is planned to incorporate lithium hydride in the boiler and sodium fluoride in a superheater and reheater. Figure 10 shows the design of the 3 kilowatt system called Sunflower-2 which is being developed by the TAPCO Group of Thompson Ramo Wooldridge in Cleveland, Ohio, for the NASA. This system is designed for potential use in spacecraft to be launched by Centaur or Saturn vehicles. It includes a 32 foot diameter petal type solar collector; a heat receiver containing lithium hydride at 1260°F around mercury boiler tubes; a combined rotating unit which carries on one shaft a three stage turbine, generator and pump; a radiator to condense mercury at 600°F; and a subcooler to provide liquid at the pump inlet under conditions of zero gravity or up to one-g acceleration in any direction. The system is designed for one year life in an earth satellite orbit and the weight is estimated to be 700 pounds. Delivery of experimental prototypes is scheduled for mid-1963. The system is to be launched in the folded configuration shown at the right in Figure 10. After injection into orbit, the collector must unfold and be oriented toward the sun within 3/4° accuracy, and then mercury is injected to bring the generator up to the design speed of 40,000 rpm.

Despite the apparent complexity of such a system, it is reassuring to know that a similar mercury turbogenerator was recently operated on the ground for 2500 hours with no indication of incipient failure.

The Stirling engine is under development by the Allison Division of General Motors for the U. S. Air Force. The high thermal efficiency of the Sterling cycle (over 30% as compared to 11% for the Sunflower mercury Rankine cycle) makes it attractive for solar powered systems in order to reduce the size and weight of the collector and thermal storage system. However, the low temperature of the cooling water (150°F to 250°F) requires a large area radiator, which is not easily packaged into a spacecraft, unless it is designed to be coiled up and to spring into a flat position in space.

The 15 kw rubidium vapor system is under development by the Sundstrand Turbo Division for the U. S. Air Force. The high turbine inlet temperature of 1750°F with a radiator temperature of 675°F will potentially yield a light weight system (819 pounds), provided that the serious materials problems for the thermal storage unit can be solved and a light weight solar collector developed. This system will require a collector diameter of about 40 feet.

d. Solar-Thermoelectric Systems

At the present time a large-scale research effort is under way in the United States on thermoelectric materials, with much of the effort supported by the U. S. Navy. Rapid progress is being made in developing properly doped semi-conductors of both the p and n type which will give their maximum conversion efficiency by the Seebeck effect at a particular temperature or over a range of temperatures. Materials such as lead telluride are commercially available which can be operated at a hot junction temperature of 500 to 650°C. Sublimation of the material in the vacuum of space will limit the life of a thermoelement unless it is properly encapsulated.

A wide variety of design concepts have been proposed for solar-thermoelectric space power systems. One concept involves a flat plate solar collector and a flat plate radiator, separated by small thermoelements at discrete intervals. The two flat plates must include electrical insulation strips to permit many elements to be connected in series. The low temperature attainable on a flat plate collector will probably limit the over-all conversion efficiency of such a system to less than 5%.

If solar concentrators are used to increase the hot junction temperature, the efficiency may be improved. The solar collectors may vary in size from a diameter of a few inches to 10 feet or more. Figure 11 shows a rather unique design for a solar-thermoelectric module which is under development by the Hamilton Standard Division of United Aircraft Corporation for the U. S. Air Force. A system would employ a large number of modules of the type shown connected in a series and parallel arrangement.

The major disadvantage of solar-thermoelectric systems is the large area of solar collector and radiator required per unit electrical output, which is due to the limited efficiency of presently available thermoelectric materials. Improved materials may make thermoelectric conversion more attractive for space applications.

e. Solar-Thermionic Systems

In the past three years, a rapidly increasing effort has been devoted to research on thermionic conversion of heat to electricity. In engineering terminology, a thermionic converter, like a thermoelectric converter, is a heat engine which uses electrons as the working fluid. The thermionic converter cathode receives heat and boils off electrons which then cross a gap to an anode where they are condensed, reject heat and then flow through an external load to deliver power before returning to the cathode. Thermionic conversion would be particularly attractive for space applications because the high cathode temperatures of 1000°C to 3000°C, or higher, permit heat rejection from the anode at temperatures of 700°C to 1000°C, resulting in small radiator areas. Power densities from 1 to 10 watts per square centimeter of cathode area are attainable.

The current density is limited by the accumulation of electron space charge in the space between cathode and anode. This effect can be reduced in a vacuum diode by decreasing the electrode spacing to 10 microns or less, but with attendant fabrication problems. A more promising solution is to provide a vapor of low ionization potential, such as cesium vapor in the interelectrode space and to generate an ionized plasma which will neutralize the electron space charge. Cesium may be ionized by surface contact ionization on the hot cathode or on a specially provided grid, or it may be ionized by electron bombardment or by other atomic collision or radiative processes within the gas volume.

Rapid progress is being made in gaining a basic understanding of the complex physical processes involved in thermionic converters. Laboratory devices have operated at thermal efficiences of 10% to 15%, with cathode temperatures as low as 1100°C to 1300°C. Relatively low cathode temperatures are particularly desirable for use with solar heat collectors in order to reduce the required accuracy of mirror fabrication and attitude control, as well as to minimize evaporation of cathode material.

Design studies indicate that solar-thermionic systems will be competitive with solar-mechanical and solar-thermoelectric systems in the power range from 1 to 20 kilowatts. However, they will require angular accuracies of the solar collector surface and of the sun-pointing control of better than 0.1 degree.

D. Nuclear Power Systems

1. Radioisotopes

The development of radioisotope heat sources with thermoelectric or thermionic conversion is technically feasible. Power outputs of a few watts may be considered, using certain isotopes which are amenable to production by reactor irradiation or by separation from fission products. Table 1 shows a list of certain isotopes with their half-lives.

The use of radioisotope units appears technically desirable for certain space missions such as lunar or planetary landing missions, where a rugged package is required, where the desire to make scientific measurements of radiation precludes the use of a nuclear reactor, or where the absence of sunlight (e.g., under the clouds of Venus or during the lunar night) prevents the use of solar energy.

The two major problems in the use of radioisotopes are the hazards during the launch phase and the possibility of radioactive contamination of the moon or planets in case of accidental impact at excessive velocity. Development tests are being conducted to determine the resistance of isotope packaging to simulated launch pad explosions and fires, impact with ground and water, and atmospheric reentry.

The NASA does not plan to utilize radioisotopes in spacecraft unless it is demonstrated that the safety and contamination problems are solved.

2. Reactor-Turboelectric Systems

An extensive literature exists on nuclear reactor-turboelectric systems, particularly with respect to their potential use with electrical space propulsion systems. Present development efforts include the development of the SNAP-2 system, which is designed for a 3 kilowatt output, and the SNAP-8, which will be capable of generating either 30 or 60 kilowatts using one or two turbo-generator systems. Both the SNAP-2 and SNAP-8 will use small reactors with zirconium-uranium hydride fuel elements, cooled by sodium-potassium alloy, with mercury Rankine cycle conversion systems. The system weights, exclusive of radiation shielding, are predicted to be 600 pounds for SNAP-2 and 1400 pounds for the 30 kw SNAP-8. The additional weight for shielding will be about 500 pounds and 900 pounds respectively for a transistorized payload, and of the order of 1000 to 2000 pounds or more for a manned vehicle, depending on the separation distance and the radiation scattering problems introduced by the specific spacecraft configuration.

For ultimate electrical propulsion applications, power levels of 300 kw to several megawatts will be desired as well as specific weights below 10 pounds per kilowatt. In order to achieve these goals, research is proceeding on high temperature alkali metal Rankine cycles with fluids such as rubidium and potassium. Turbine inlet temperatures of 1600°F to 2000°F may be possible with fast reactors utilizing fuel elements made of ceramic, such as uranium monocarbide, and clad with refractory metals.

The use of Brayton cycle gas turbine conversion systems will not be competitive with the Rankine cycle for high power nuclear-electric systems, because of the large radiator area requirements, even with gas turbine inlet temperatures above 3000°F.

The present concept for operational use of a nuclear-electric system involves start-up of the reactor after injection into a high orbit. By this technique, no hazardous fission products would be present during the launch phase. If the orbit is low enough that reentry would eventually occur, provision would be made to assure reactor burn-up at high altitude during reentry.

3. Reactor Thermoelectric Systems

For small nuclear reactors with electrical output in the range of 300 watts to 3 kilowatts, the use of semiconductor thermoelements instead of a mechanical conversion system appears to be superior with respect to low weight, simplicity, reliability and absence of moving parts. The Atomic Energy Commission is developing a 300 watt unit called SNAP-10, which will weigh about 350 pounds, excluding shielding. For power levels above one kilowatt, a heat transfer loop (e.g., Na-K) may be used to transfer energy from the reactor to a separate thermoelectric converter and radiator unit.

Optimum thermoelectric materials are generally doped to produce the high impurity concentrations of about 10^{19} atoms per cubic centimeter. Consequently, they can tolerate higher radiation exposures than solar cells or transistors, which have lower doping concentrations of the order of 10^{16} atoms per cubic centimeter.

The development of improved thermoelectric materials may extend the feasibility of reactor-thermoelectric systems up to tens of kilowatts.

4. Reactor-Thermionic Systems

The use of thermionic conversion instead of turboelectric conversion appears very promising, particularly for the high power range of several megawatts, which will be required for useful electrical propulsion missions. Primary emphasis is being placed on the development of a rod-type fuel element, surrounded by an annular cesium vapor diode, with a cooling loop to transfer heat from the anodes to an external radiator. A major effort is required to develop a stable fuel element for long time operation at a temperature above 1500°C. Problems include: Dimensional stability of the fuel, diffusion and disposal of fission products, cladding of the fuel with a suitable cathode surface, and electrical insulation of series-connected diodes from a liquid metal anode coolant. Design studies indicate that such a thermionic reactor may weigh only 5 pounds per kilowatt output for power levels of several megawatts.

Very little attention is being given to systems having the thermionic converter external to the reactor, because of the difficulty of operating a heat transfer fluid loop at temperatures above 1500°C.

IV. CRITERIA FOR POWER SYSTEM SELECTION

The selection of a particular type of power supply for a specific spacecraft is a difficult task which involves evaluation of various possible power systems with respect to the following criteria:

A. Development status and availability to meet desired schedule and to operate with sufficient reliability.

B. Weight.

C. Compatibility with spacecraft design requirements, such as allowable volume, tolerance of the payload to temperature or nuclear radiation, and compatibility with the attitude orientation requirements of the spacecraft and its sensors and antennas with the geometry of the power supply.

D. The degree to which any moving parts of the power system produce undesirable perturbing torques which must be compensated by the attitude control system (e.g., friction in bearings supporting an oriented solar collector, or gyro torques caused by precession or imperfect speed control of a rotating generator).

E. Tolerance of the power system for the environment imposed by the launch vehicle and spacecraft, such as accelerations, vibrations, and temperatures.

F. Resistance of the power system to effects of the space environment, such as meteoroid hits, high vacuum and solar radiation effects on materials, radiation damage by trapped radiation on components (such as solar cells or plastics) and the degree of sensitivity of the power system to variation of the environment from sun to shade or with a varying distance from the sun.

G. Radioactivity hazards and contamination problems, for nuclear systems.

H. Degree to which the power system can be integrated with the spacecraft environmental control system (e.g., the use of chemical fuels to cool the spacecraft).

I. Costs of power system development and production.

It is recognized that a really adequate comparison of various systems with respect to all these criteria cannot be made until various components and sub-systems are developed and tested in simulated or actual space environments. A selection of the types and sizes of power systems which should be developed for feasibility demonstration must be based upon a knowledge of the various spacecraft requirements, together with engineering judgment of the relative development potential of various power systems. It is believed that attention should be devoted to the prediction of system reliability, during the process of system selection and preliminary design.

V. DISCUSSION AND CONCLUSIONS

Power system weight is an important factor which can be estimated quantitatively by engineering analysis, and a number of weight comparisons have been published. Figure 12 shows the power system weight in pounds as a function of power output in kilowatts (adapted in part from estimates by Slone and Lieblein of the NASA Lewis Research Center). It is seen that in various power ranges, consideration should be given to the systems indicated in Table 2.

For the power range below 300 watts, solar photovolaic systems appear most promising, with solar-heated thermoelectric and thermionic systems offering the possibility of lower production cost but with the requirement of more accurate sun orientation. Radioisotopes may be considered for special mission requirements, assuming that hazards and contamination problems can be solved, and that the desired isotope can be produced in sufficient quantity.

For the power range of 300 watts to 5 kilowatts, at least seven types of systems are competitive with respect to weight. In this range, there are incentives to develop solar-thermoelectric or thermionic systems instead of solar cells in order to avoid sensitivity to Van Allen radiation and to reduce costs. The reactor-thermoelectric system (e.g., the 300 watt SNAP-10) is desirable for simplicity, reliability and absence of sun-orientation requirements. At the 3 kw level, the solar, Rankine cycle turboelectric system (Sunflower-I) and the solar-Stirling cycle reciprocating engine system are competitive with the reactor Rankine cycle turboelectric system (SNAP-2). Future development (5 to 6 years) will probably result in solar-thermionic systems with thermal energy storage which will replace the mechanical conversion systems in the power range of a few kilowatts.

For the power range of 5 kw to 30 kw, turboelectric and possibly thermionic conversion systems will be desirable. Solar energy will be preferable for multi-man space stations because its use avoids the shielding weight penalty of a reactor, and the hazards of reactor reentry from the low orbital altitudes (underneath the inner Van Allen belt) which are required for long duration manned satellites.

For initial experimental missions with electrical propulsion, the 30 kw or 60 kw SNA_-8 will be developed. Higher nuclear power systems will utilize alkali metal Rankine cycles (e.g., potassium) and eventually thermionic conversion in order to reach the low specific weights which are required to make electrical propulsion competitive with nuclear-heated and chemical rocket propulsion.

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TABLE I

RADIOISOTOPES FOR POSSIBLE USE IN SPACE POWER UNITS

	Isotope	Radiation	Half-Life
1.	Po-210	Alpha	138.4 days
2.	Cm-242	Alpha	162.5 days
3.	Ce-144	Beta and Gamma	285 days
4.	Pm-147	Beta	2.6 years
5.	Sr-90	Beta	27.7 years
6.	Pu-238	Alpha	86.4 years

TABLE 2

LONG DURATION POWER SYSTEMS FOR VARIOUS POWER RANGES

A. Below 300 w

1. Solar cells with batteries
2. Solar-thermoelectric
3. Solar-thermionic
4. Isotope-thermoelectric
5. Isotope-thermionic

B. 300 w to 5 kw

1. Solar cells with batteries
2. Solar-thermoelectric
3. Solar-thermionic
4. Reactor-thermoelectric
5. Solar-turboelectric
6. Solar-Stirling cycle
7. Reactor-turboelectric

C. 5 kw to 30 kw:

1. Solar-turboelectric
2. Solar-thermionic
3. Reactor-turboelectric
4. Reactor-thermionic

D. Above 30 kw

1. Reactor-turboelectric
2. Reactor-thermionic

ESTIMATED ELECTRIC POWER REQUIREMENTS

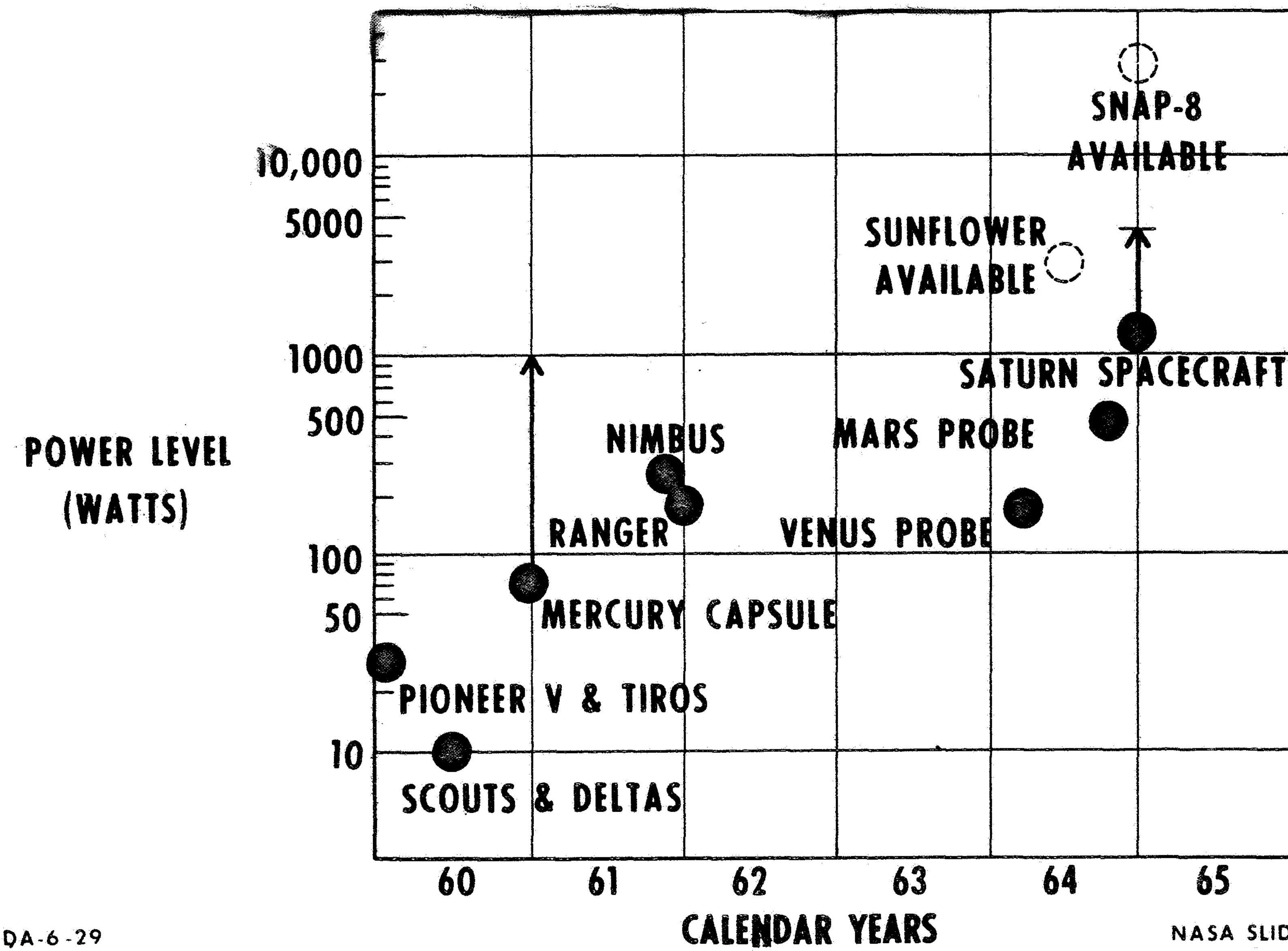

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FIG. 1

ENERGY SOURCES & CONVERSION METHODS

CONVERSION SYSTEM	ENERGY SOURCE			
	CHEMICAL	SOLAR RADIATION (OR CHARGED PARTICLES)	NUCLEAR REACTOR	RADIOISOTOPE
OPEN CYCLE TURBINE	○	N. A.	N. A.	N. A.
CLOSED CYCLE TURBINE	◎	◎	◎	◎
~~CLOSED~~ OPEN CYCLE RECIPROCATING ENGINE	◎	N. A.	N. A.	N. A.
CLOSED CYCLE RECIPROCATING ENGINE (STIRLING CYCLE)	●	◎	●	N. A.
OPEN CYCLE MFD CONVERTER	◎	N. A.	N. A.	N. A.
CLOSED CYCLE MFD CONVERTER	●	●	◎	N. A.
BATTERY	○	N. A.	N. A.	N. A.
OPEN CYCLE FUEL CELL	◎	N. A.	N. A.	N. A.
THERMALLY REGENERATIVE FUEL CELL	N. A.	◎	◎	●
PHOTO. OR RADIO-REGENERATIVE FUEL CELL	N. A.	◎	●	◎
THERMIONIC CONVERTER	◎	◎	◎	◎
THERMOELECTRIC CONVERTER	◎	◎	◎	○
PHOTOVOLTAIC P-N JUNCTION	N. A.	○	N. A.	N. A.
CHARGED PARTICLE ACTIVATION OF P-N JUNCTION	N. A.	●	●	●
ELECTROSTATIC POWER GENERATION BY CHARGED PARTICLES	N. A.	●	◎	○
ELECTRON EMISSION BY PHOTONS OR CHARGED PARTICLES	N. A.	◎	●	●
FERROELECTRIC	N. A.	●	●	●
THERMOMAGNETIC	N. A.	●	●	●
PYROELECTRIC	N. A.	●	●	●

○ DEVELOPED & AVAILABLE ◎ UNDER DEVELOPMENT ● NOT ADEQUATELY INVESTIGATED N.A. NOT APPLICABLE

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FIG. 2

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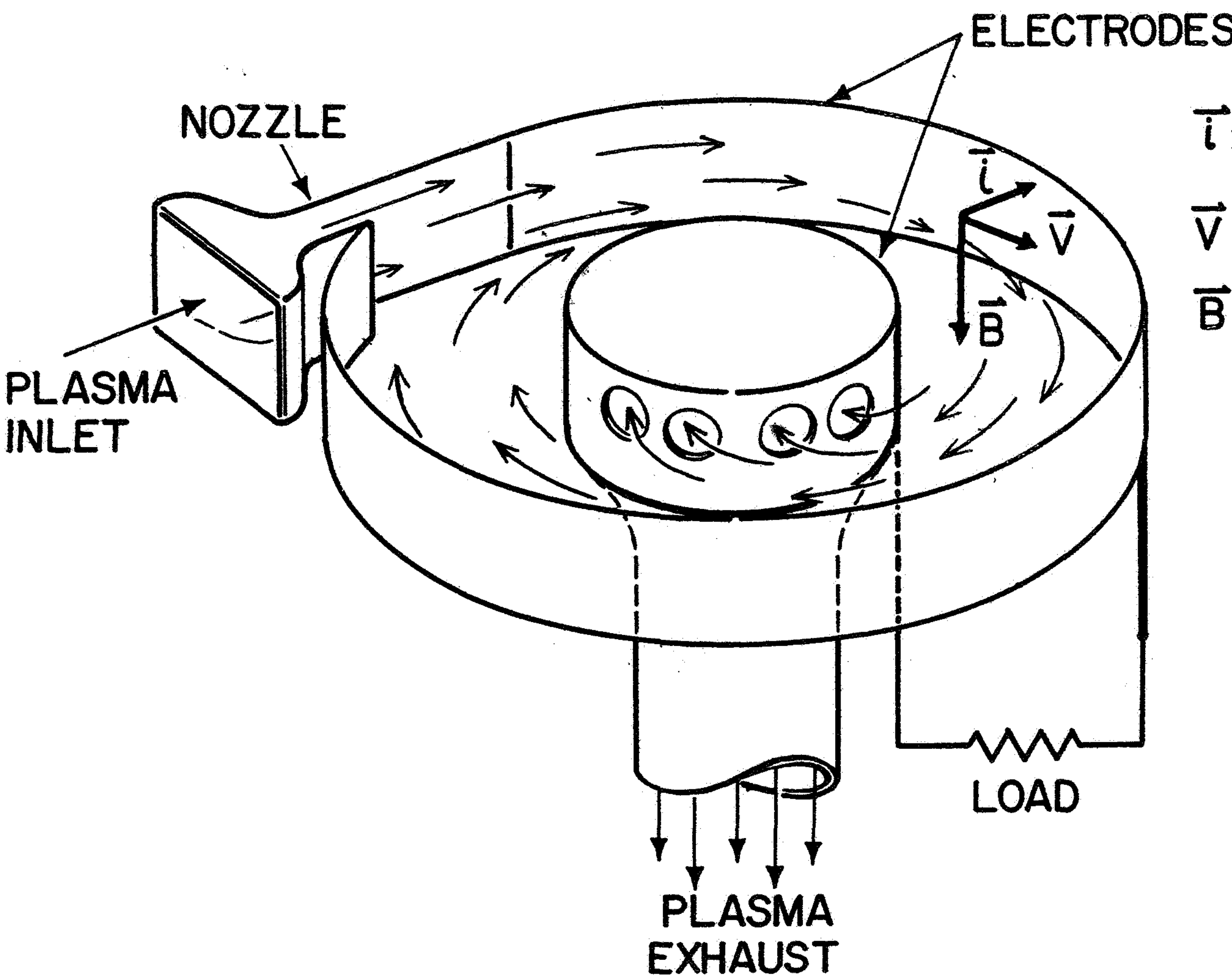
SCHEMATIC CONCEPT
OF A VORTEX MAGNETO-FLUID-DYNAMIC GENERATOR
(Magnet to produce axial magnetic field, $\vec{B}$ is not shown)



$\vec{i}$ = CURRENT

$\vec{V}$ = PLASMA VELOCITY

$\vec{B}$ = MAGNETIC FIELD

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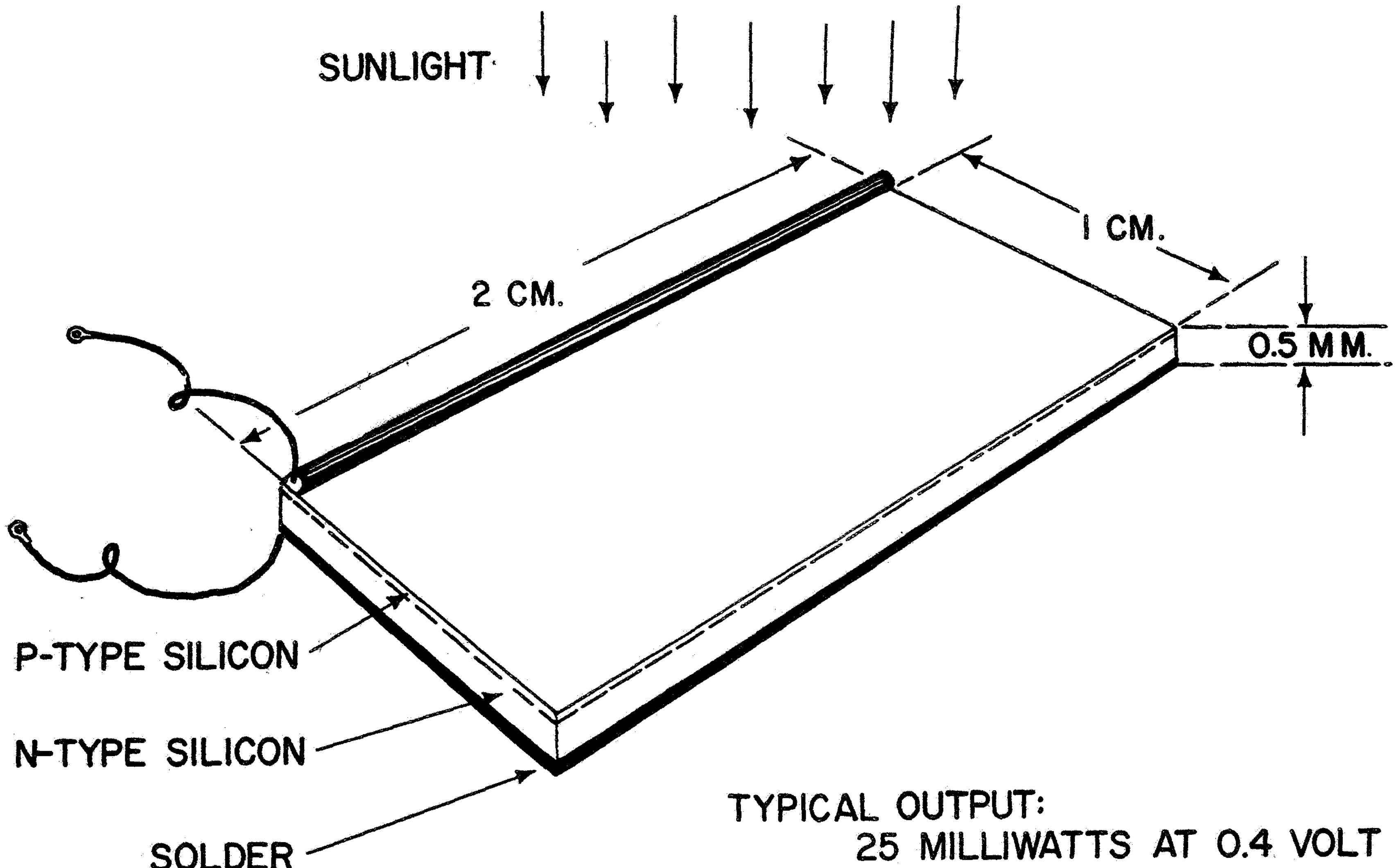
FIG. 3

SKETCH OF A TYPICAL PHOTOVOLTAIC SOLAR CELL



TYPICAL OUTPUT:
25 MILLIWATTS AT 0.4 VOLT
FOR 10 % EFFICIENCY

FIG. 4

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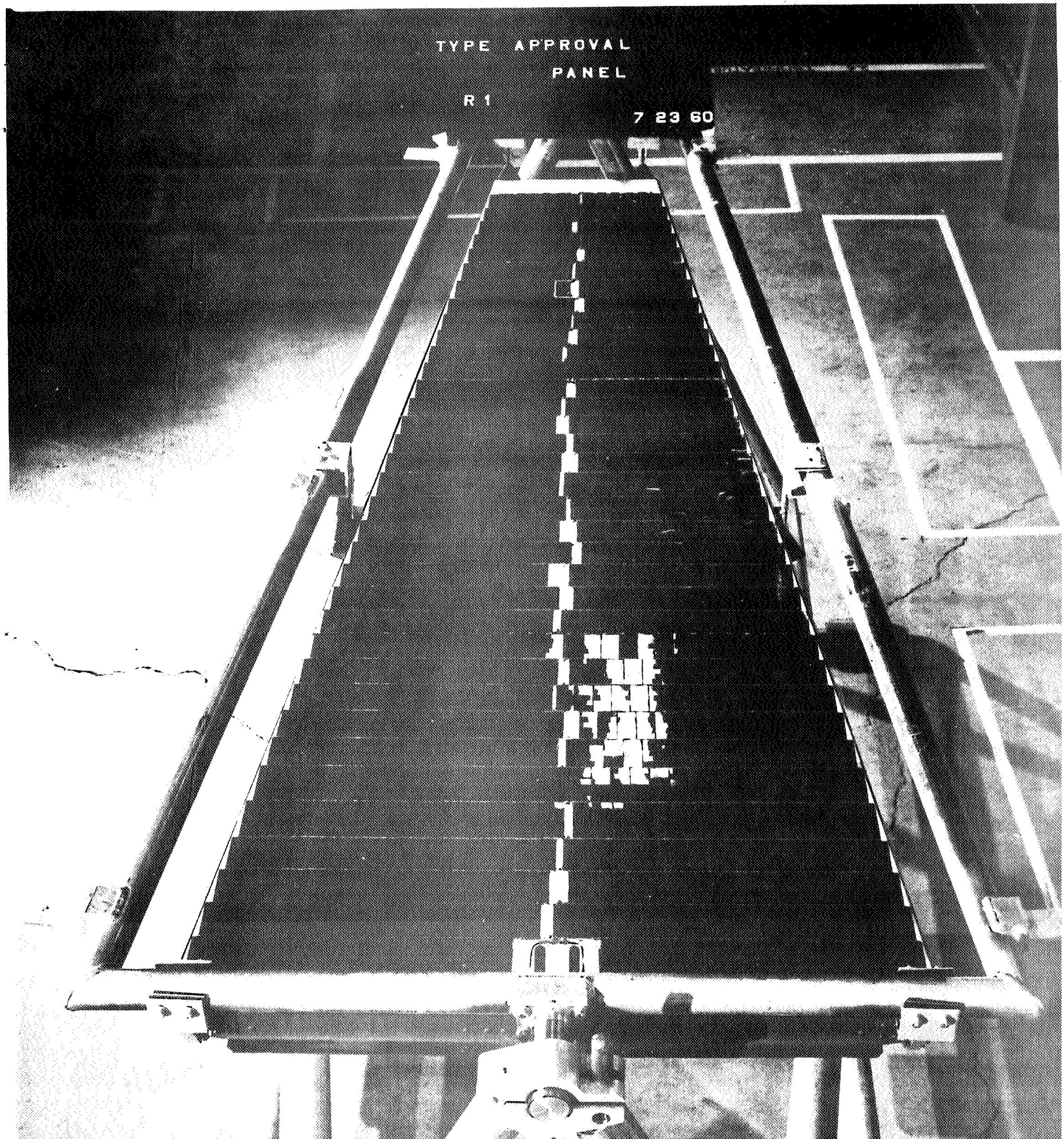
NASA SLIDE 2762



10 SQUARE FOOT SOLAR CELL PANEL FOR RANGER SPACECRAFT
(Jet Propulsion Laboratory)
FIG. 5

10 FOOT UMBRELLA-TYPE SOLAR COLLECTOR (LANGLEY RESEARCH CENTER)

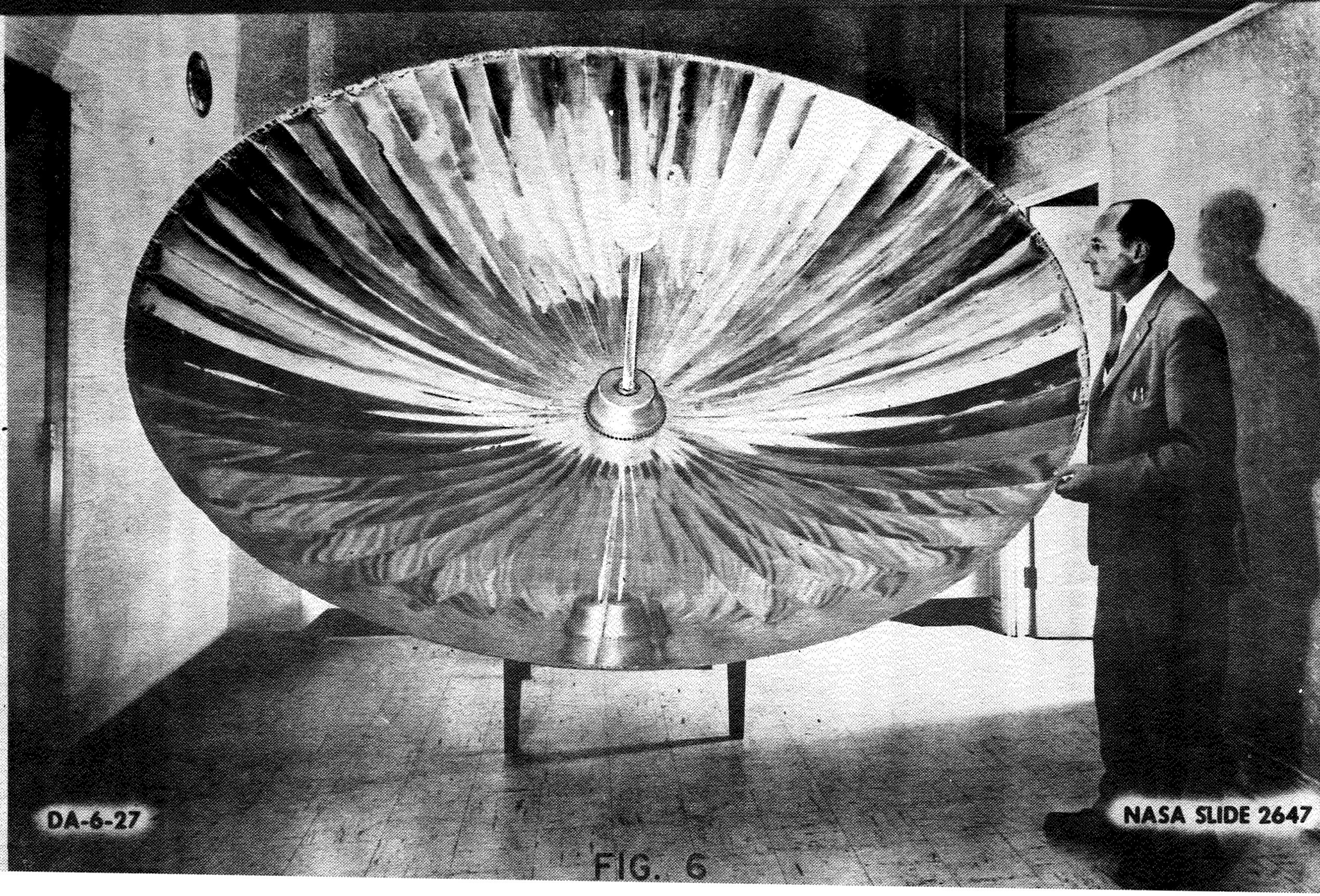
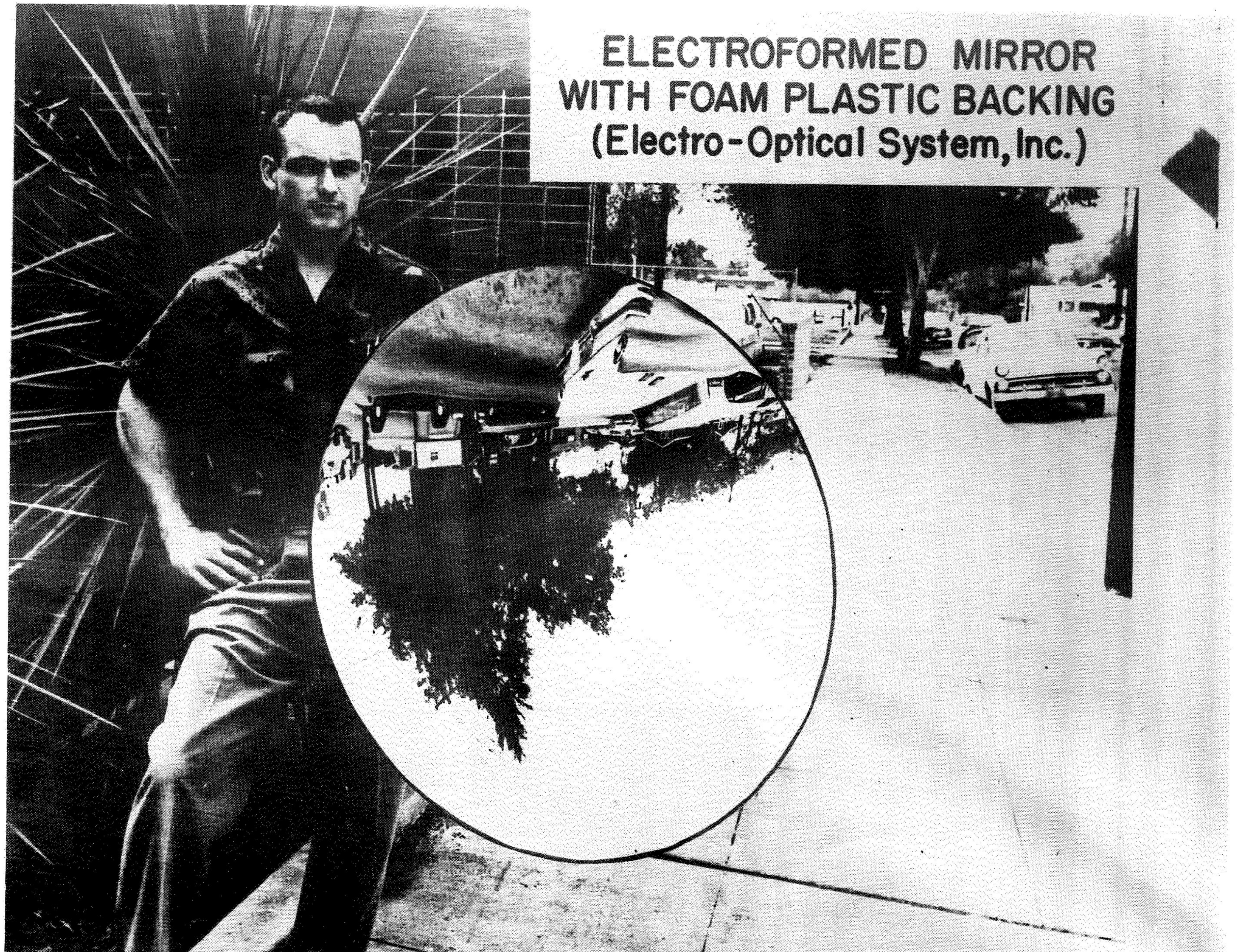
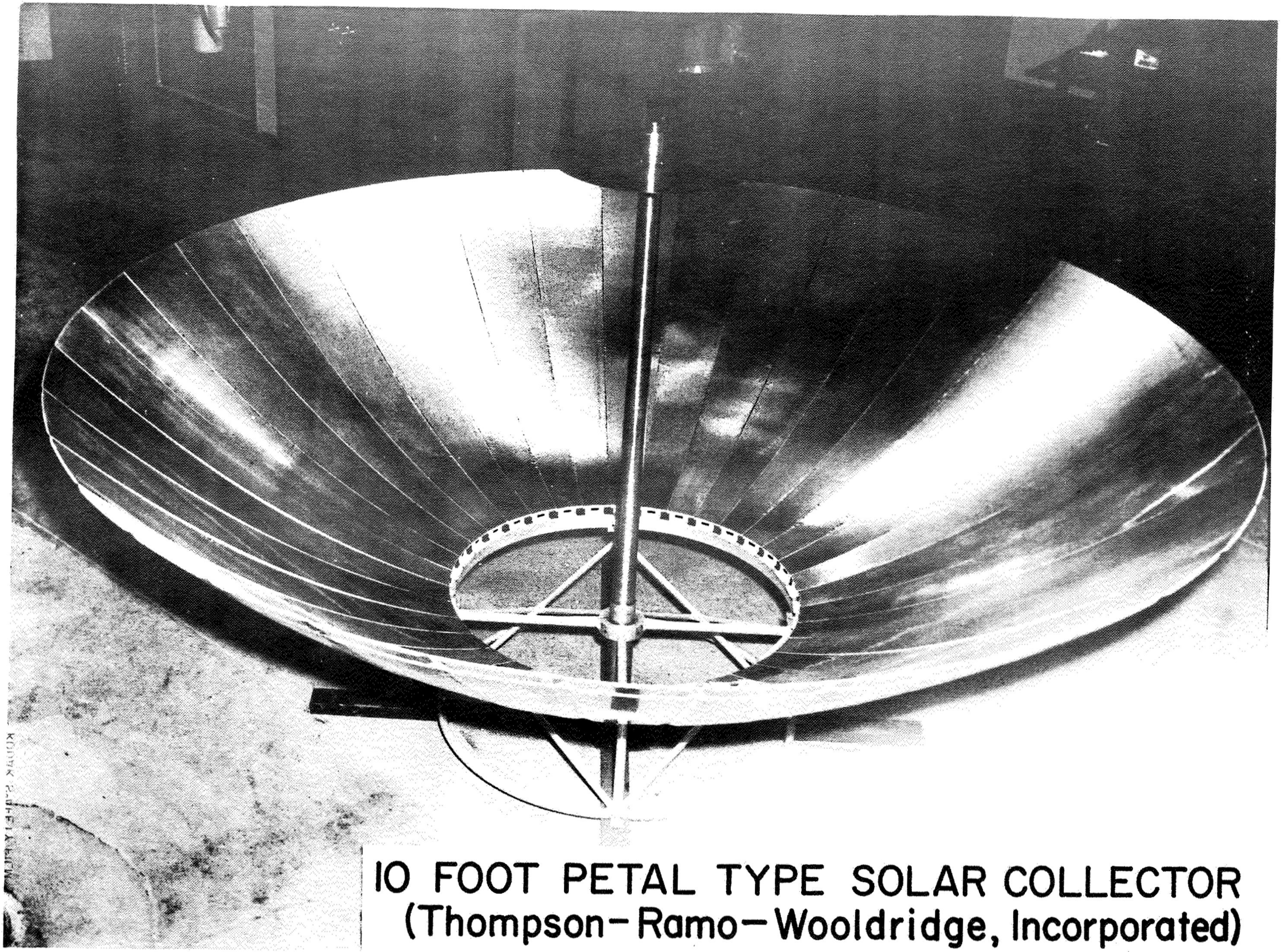


FIG. 6



ELECTROFORMED MIRROR
WITH FOAM PLASTIC BACKING
(Electro-Optical System, Inc.)

FIG. 7



10 FOOT PETAL TYPE SOLAR COLLECTOR
(Thompson-Ramo-Wooldridge, Incorporated)

FIG 8

HEAT OF FUSION vs MELTING POINT FOR THERMAL STORAGE MATERIALS TO BE USED WITH SOLAR-HEATED SYSTEMS

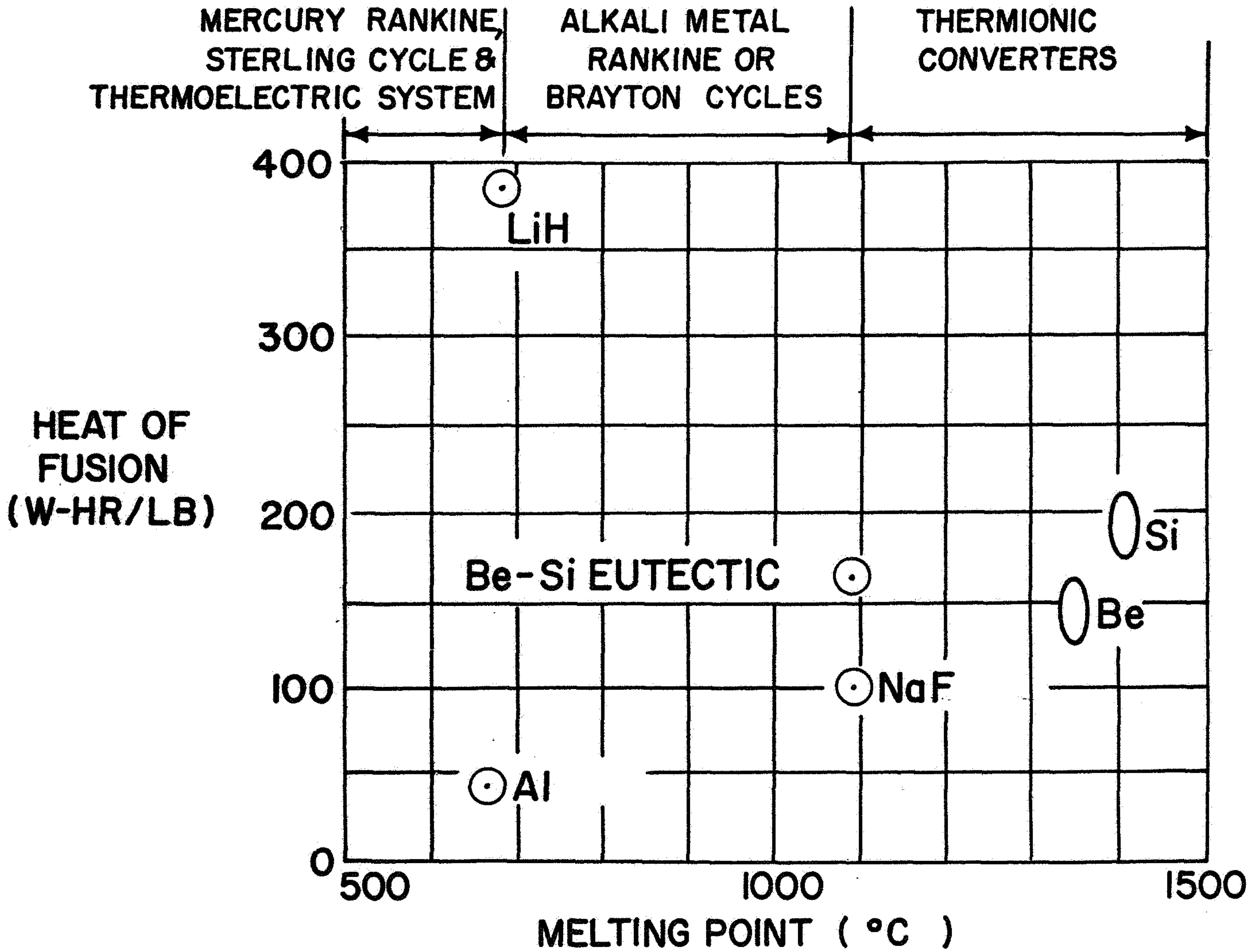


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FIG. 9

SUNFLOWER 1 SOLAR AUXILIARY POWER SYSTEM

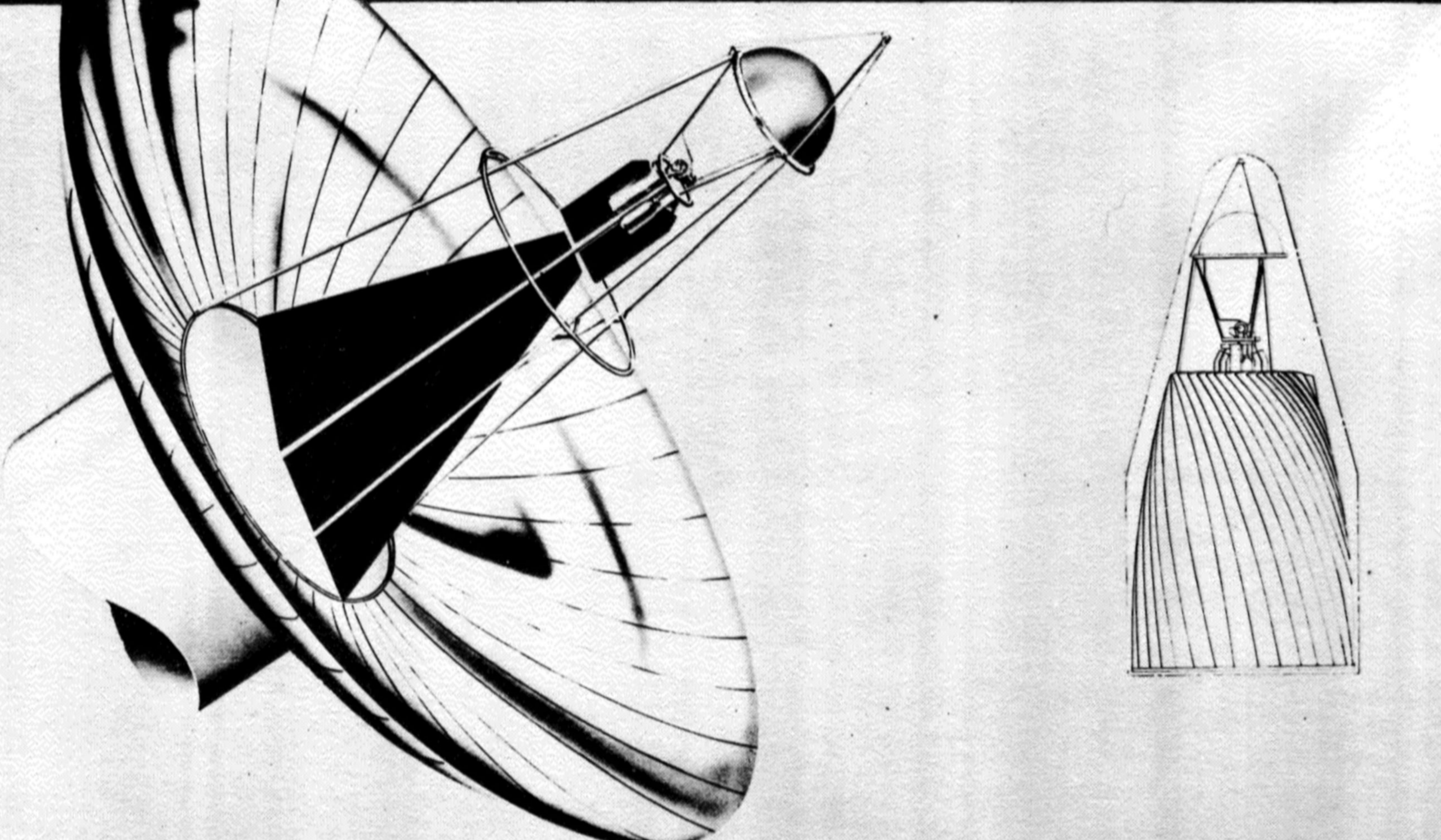

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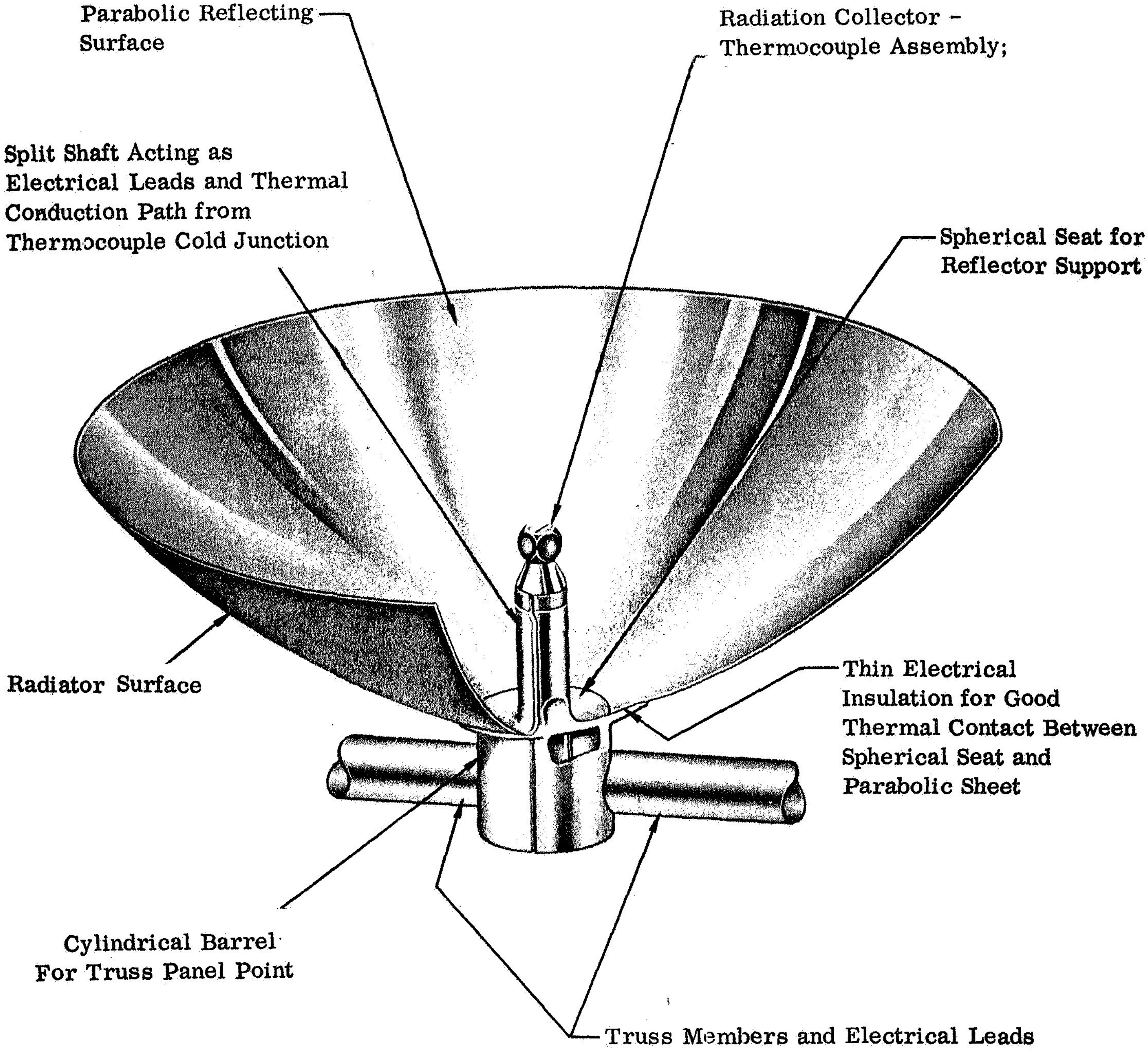
FIG. 10



MODULAR SOLAR-THERMOELECTRIC UNIT (UNDER DEVELOPMENT BY HAMILTON STANDARD FOR U.S. AIR FORCE)

FIG. 11

ESTIMATED WEIGHTS OF SPACE POWER SYSTEMS

(SOLAR POWER SUPPLIES INCLUDE STORAGE FOR LOW ORBIT SATELLITE)

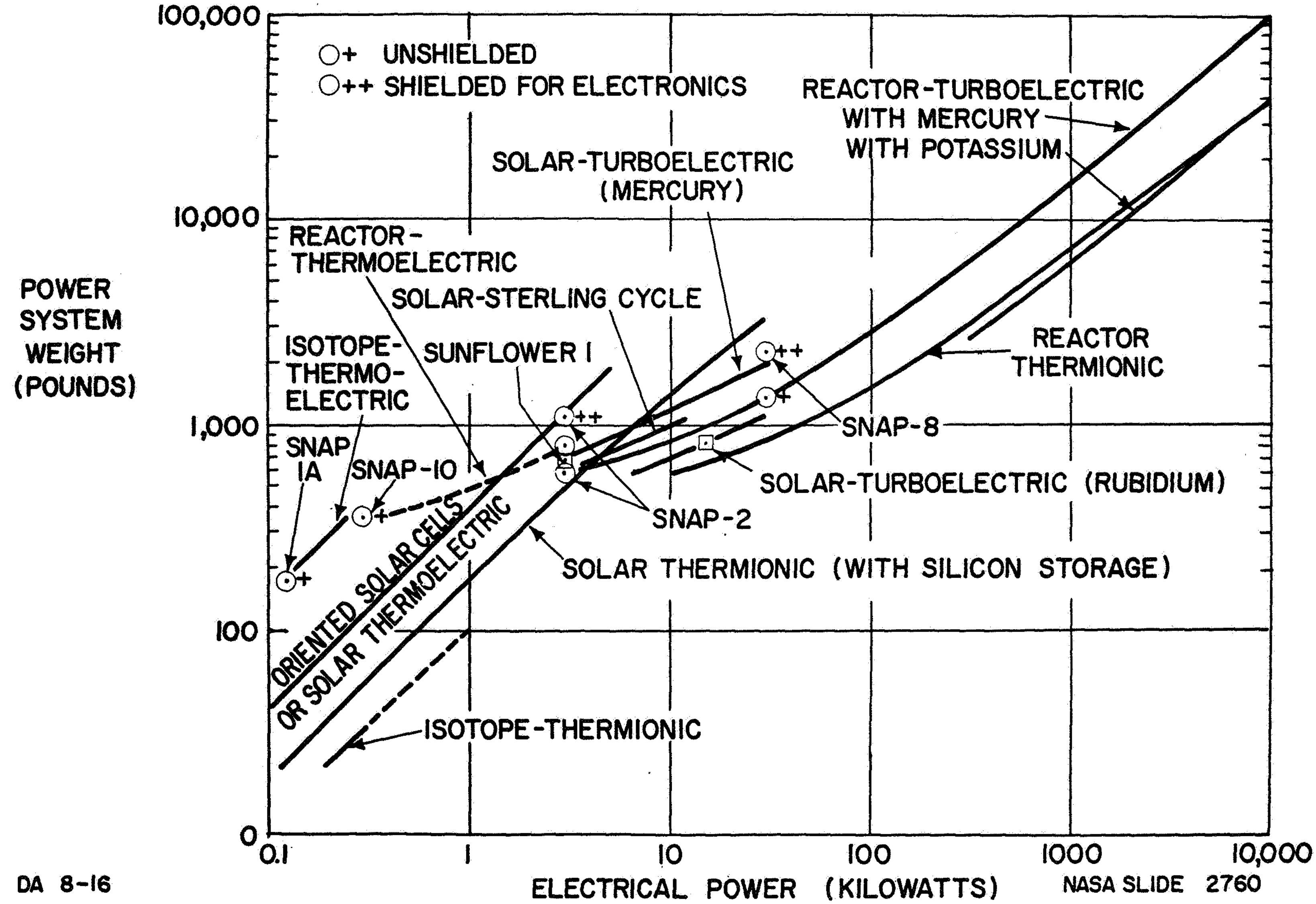


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FIG. 12